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RESEARCH MEMORANDUM

THRUST AUGMENTATION OF A TURBOJET ENGINE BY THE INTRODUCTION OF LIQUID AMMONIA INTO THE COMPRESSOR INLET

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NATIONAL ADVISORY COMMITTEE
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2608

SUMMARY

An experimental investigation was conducted to determine the thrust augmentation possible by the injection of liquid ammonia into the compressor inlet of an axial-flow-type turbojet engine. The flight conditions simulated were an altitude of 35,000 feet and a Mach number of 1.0. Ram air temperatures of 80° F and -40° F were used in addition to the standard ram air temperature of 13° F.

The maximum net thrust augmentation obtained at the standard inlet (ram) air temperature was 13 percent at an ammonia-air ratio of 0.049 pound of ammonia per pound of air, corresponding to an augmented liquid ratio of 3.8. With 80° F inlet-air temperature, 29 percent augmentation was obtained at an ammonia-air ratio of 0.055 pound of ammonia per pound of air, corresponding to an augmented liquid ratio of 4.5.

INTRODUCTION

In order to provide a method of fulfilling the requirement of additional turbojet engine thrust necessary for take-off, transonic acceleration, and combat emergency conditions, the NACA Lewis laboratory has been engaged in a general thrust-augmentation program. In the sea-level, zero-ram phase of this program, it was demonstrated (reference 1) that by the combination of the two basic methods of thrust augmentation, namely compressor coolant injection and afterburning, greater amounts of augmentation are available than can be provided by either method individually.

The initial phase of the current altitude augmentation investigation included the selection of a suitable coolant for compressor injection which eventually could be used in conjunction with afterburning. Water-alcohol mixtures provided reasonable thrust increases at sea-level, zero-ram conditions (reference 2); but when this coolant was employed in one engine at altitude flight conditions, the water-alcohol mixture was found to be an unsatisfactory coolant, providing a very limited amount of thrust augmentation (reference 3). The principle of coolant injection as a means of thrust augmentation was still considered sound for flight operation, but the need of a more suitable coolant was recognized. The high heat of vaporization and the relatively low boiling point of liquid anhydrous ammonia suggested its use as a coolant at the temperatures encountered with high-altitude flight. It should be recognized that the water-alcohol coolant presented certain limitations when used in combination with afterburning through the reduction of combustion temperature and efficiency (reference 1). These limitations may be less severe or completely absent when ammonia is used, since it is believed that the ammonia, which is combustible, would have a less deleterious influence on combustion.

Accordingly, an investigation was conducted to determine experimentally the thrust augmentation available at high-altitude flight conditions with the introduction of liquid ammonia into the compressor inlet (ammonia-air ratios up to 0.055) of an axial-flow-type turbojet engine operating at a simulated altitude of 35,000 feet and a flight Mach number of 1.0. Resultant changes in engine-performance parameters are presented, in addition to the thrust augmentation achieved and the accompanying coolant-air ratios required.

APPARATUS AND PROCEDURE

Engine

The axial-flow turbojet engine used in this investigation (fig. 1) developed 3000 pounds thrust at sea-level, zero-ram conditions, with an average turbine-outlet temperature of 1610° R and a rated speed of 12,500 rpm. The fuel used was clear unleaded gasoline.

The primary engine components were an 11-stage axial-flow compressor, a compressor-outlet mixer, a double-annular combustor of the through-flow type, a two-stage axial-flow turbine, and a variable-area exhaust nozzle. The compressor-outlet mixer produced a more favorable compressor-outlet velocity profile and consequently an improved turbine-outlet radial-temperature distribution. NACA compressor blade-clearance indicators (described in reference 3) were installed at the seventh, eighth, ninth, and tenth compressor stages. These indicators warn of impending blade-rubbing that may occur when the compressor casing is overcooled as a result of the centrifugence of unvaporized coolant. The engine was mounted in an altitude test chamber as shown in figure 2. A photograph of the engine installed in the test chamber is presented in figure 3.

2608

Ammonia-Injection System

One manifold (fig.4) containing 10 spray tubes and 10 conical spray nozzles was used to inject ammonia into the inlet-air stream approximately 68 inches upstream of the compressor inlet (fig. 1). The capacity of the cônical spray nozzles was 40 gallons per hour. The spray tubes were 1/4 inch in diameter, each containing four holes of 0.0287-inch diameter. The conical spray nozzles sprayed over the inner one-third of the inlet area, and the spray tubes distributed their flow over the outer two-thirds of the inlet area. The conical nozzles sprayed downstream; the spray bars, perpendicular to the stream. The manifold was divided so that the ammonia flow to the nozzles was independent of the flow to the spray tubes, and individual control of the nozzles and spray tubes was possible.

A schematic drawing of the ammonia-injection system is presented in figure 5. Ammonia was supplied to the engine from a storage tank, pressurized to approximately 400 pounds per square inch gage with high-pressure helium. Ammonia flow to the engine was controlled by manually operated throttle valves.

Instrumentation

Pressure and temperatures throughout the engine were measured at stations designated in figure 1. Survey probes were placed on centers of equal areas of each cross section. Instrumentation at stations 1, 3, 5, and 8 was as shown in figure 6. Aspirating-type thermocouples were placed at three locations between the injection manifold and the compressor inlet. Exhaust pressure in the altitude test chamber was measured in the plane of the exhaust-nozzle exit. Fuel flow was measured by means of rotameters; ammonia flow, by the use of a sharp-edge orifice and a differential manometer.

Procedure

The flight conditions simulated were an altitude of 35,000 feet and a Mach number of 1.0. Performance was investigated at inlet-air total temperatures of 80°, 13° (NACA standard for this flight condition), and -40° F in order to ascertain the effect of inlet-air temperature on thrust augmentation. The engine was operated at rated conditions, a speed of 12,500 rpm and an average turbine-outlet temperature of 1610° R. Turbine-outlet temperature was maintained at this value for all runs by adjusting the variable-area exhaust nozzle. By use of instrumentation at station 5, it was possible to determine the radial temperature distribution at the turbine outlet.

Ammonia was injected into the engine inlet over the following range of ammonia-air ratios:

Inlet temperature (°F)	Ammonia-air ratios
80	0 - 0.055
13	0 - .049
-40	0 - .046

2608

Normal performance runs were made at the same engine and flight conditions as the augmented performance runs in order to provide a basis of comparison for the augmented performance. Normal performance was determined with the ammonia-injection system installed in the engine. Thrust and air flow were calculated as described in the appendix.

RESULTS AND DISCUSSION

Selection of Coolant

The cooling that results from the evaporation of liquid both before and during compression of the air passing through a turbojet engine has been demonstrated to be a suitable means of thrust augmentation. Early studies in which water and water-alcohol mixtures were used at sea-level, zero-ram conditions led to the use of a water-alcohol mixture at simulated high-altitude flight conditions. The water-alcohol mixture proved to be an unsatisfactory coolant in one engine investigated at a high altitude (reference 3), and the need for a more favorable coolant was recognized. A number of liquids were considered as internal coolants for use at temperatures from 13° F to 80° F. A compilation of the more favorable liquid coolants considered is presented in the following table:

Coolant	Heat of vaporization (Btu/lb)	Boiling point at atmospheric pressure (°F)
Water	1079	212
Hydrogen peroxide	587	306
Liquid ammonia	561	-27
Methyl chloride	180	-109
Liquid air	93	-320

Water and hydrogen peroxide were disqualified as satisfactory coolants at the relatively low operating temperature, since their boiling temperatures are relatively high. Methyl chloride and liquid air were considered unsuitable because their heats of vaporization are comparatively low, and thus these coolants would absorb only small quantities of heat during evaporation.

Liquid anhydrous ammonia was therefore selected as the coolant
2608 possessing the greatest advantage for this application, since it has a boiling temperature sufficiently low to permit favorable evaporation at the temperatures and pressures under consideration. Also, ammonia will burn, thereby replacing some of the engine fuel. It has a heat of combustion of about half that of gasoline, and flammable mixtures are formed at ammonia-air ratios between 0.095 and 0.15 pound of ammonia per pound of air. The freezing temperature of ammonia is -108° F; the boiling temperature varies with absolute pressure as shown in figure 7. The density of gaseous ammonia is approximately one-half that of air at comparable temperatures. The principal disadvantages of ammonia are that it is toxic and will attack copper and copper alloys in the presence of moisture.

Cooling with Liquid Ammonia

Thorough mixing of the ammonia and air at the engine inlet was found to be of primary importance in the achievement of the maximum thrust augmentation possible with the quantity of ammonia being evaporated. Preliminary tests in which spring-loaded spray nozzles were used to inject the ammonia resulted in a poor circumferential distribution and produced only 50 percent of the thrust increase that was possible with the final improved spray configuration. The spring-loaded nozzles were removed and replaced by fixed conical spray nozzles which improved the circumferential distribution but provided a relatively poor radial distribution. The use of this configuration produced only 80 percent of the thrust increase ultimately obtained. The final spray configuration (fig. 4), incorporating the experience of preliminary designs, provided a more uniform distribution of the ammonia and air and enabled attainment of the maximum amount of thrust augmentation. Best results were obtained when one-third of the total ammonia flow was injected through the conical spray nozzles and the remaining two-thirds through the spray bars.

The liquid ammonia would be expected to vaporize rapidly at temperatures between 13° F and 80° F. A study of the temperatures measured with several aspirating-type thermocouples between the injection station and the compressor inlet (fig. 8) indicated that the ammonia was completely evaporated in a distance of not more than 3 feet. Air velocity at the injection station varied from 300 to 350 feet per second over the range of inlet temperatures and injection rates investigated.

The decrease in compressor-outlet temperature caused by the evaporation of ammonia is shown in figure 9. The temperature drop at the compressor inlet will be approximately equal to the drop at the compressor outlet until the condition is reached at which the inlet air is saturated with ammonia. At that point, the inlet temperature can no longer decrease. Calculations indicate that an ammonia-air ratio of 0.06 pound of ammonia per pound of air will not saturate the inlet until temperatures below -100° F are reached. Conditions of saturation were never reached during this investigation, and the temperature drop at the compressor inlet was generally within ±2° of the temperature drop at the compressor outlet. 2608

Engine Performance

Augmented net thrust ratio versus augmented liquid ratio obtained at an altitude of 35,000 feet and flight Mach number of 1.0 for rated engine conditions is plotted in figure 10. Augmented liquid ratio is defined as the sum of the ammonia flow and fuel flow divided by the normal engine fuel flow. Maximum thrust augmentation at the 80° F inlet temperature was 29 percent at an augmented liquid ratio of 4.5; at the standard inlet temperature of 13° F it was 13 percent at an augmented liquid ratio of 3.8; and at -40° F inlet temperature it was 9 percent at an augmented liquid ratio of 3.4. For comparison, the augmentation reported in reference 3 for interstage water-alcohol injection at inlet-air temperatures of 80° and 13° F is shown by the two dashed curves. Maximum augmentation with water-alcohol injection was 10.8 percent with an 80° F inlet-air temperature and 5.5 percent with a 13° F inlet-air temperature.

Thrust values with ammonia injection are presented in figure 11. At an ammonia-air ratio of approximately 0.05, there was a net thrust increase (fig. 11(a)) of 292 pounds with an 80° F inlet-air temperature, and an increase of 167 pounds with a 13° F inlet-air temperature. At -40° F inlet-air temperature, the ammonia-air ratio was limited to about 0.046 because of engine blow-out. At the two higher inlet-air temperatures, no difficulties were encountered which would prevent using even higher ammonia flows, resulting in greater thrust gains. However, in order to prevent ignition of the mixture before it enters the combustion chamber, the maximum flow should probably be limited to an ammonia-air ratio (by weight) of less than 0.095, since a combustible mixture is formed at that ratio.

Jet thrust is presented in figure 11(b) along with a jet thrust calculated from engine performance characteristics based solely on the drop in inlet-air temperature caused by evaporation of the ammonia (see appendix). Changes in density of the inlet mixture from that of 100 percent air, caused by the presence of ammonia vapor, were accounted for in the computation; changes in the specific heat ratio were small and were therefore neglected. The values of jet thrust obtained (fig. 11(b)) at the high ammonia-air ratios were 1 to 3 percent below the calculated values. This difference between experimental and calculated thrust can be explained by an examination of the inlet weight 2608 flow (air flow plus ammonia flow) and compressor-pressure-ratio characteristics presented in figures 12 and 13.

Inlet weight flow as a function of ammonia-air ratio is presented in figure 12 for the three inlet-air temperatures investigated. Throughout the entire range of ammonia-air ratios the measured weight flow was less than that calculated. The difference was from 1 to 2 percent at the high ammonia-air ratios. This difference is attributed to a nonuniform temperature distribution at the engine inlet. It will be noted that at 13° F inlet-air temperature, the separation between measured and calculated weight flow was considerably greater than that for the other inlet-air temperatures. Examination of the aspirating-type thermocouple readings indicated large temperature gradients (45° F) which were less pronounced at the other inlet-air temperatures. Consequently, a further improvement in distribution of ammonia at the engine inlet may provide a slight additional increase in weight flow with a proportionate increase in thrust.

As noted in figure 12, the inlet weight flow for the 80° F temperature was increased 3.1 pounds per second by injecting 0.055 pound of ammonia per pound of air. The attendant increase in air flow amounted to 1.6 pounds per second, or about half of the total increase in inlet weight flow. With the 13° inlet-air temperature the inlet weight flow increased 1.8 pounds per second of which 1.4 were ammonia and 0.4 was air. Air-flow data at -40° F with injection were unreliable because of frozen inlet-pressure tubes. The dotted line for -40° F was based on the normal-performance air flow, which was assumed to remain constant with ammonia injection; therefore, the increase in weight flow shown for -40° F was due solely to the addition of the ammonia.

As shown in figure 13, the compressor pressure ratio increased with ammonia injection for all inlet-air temperatures investigated, the increase being more pronounced at the higher inlet-air temperatures. Also, the separation between experimental and calculated performance increased at the higher inlet-air temperatures. The deviation of the experimental values from the calculated values of compressor pressure ratio and inlet total weight flow therefore accounts for the thrust deviation from the calculated value which was shown in figure 11(b).

The variation in fuel flow and ammonia flow with increasing ammonia-air ratio is presented in figure 14. There was a reduction in fuel flow as ammonia flow was increased, amounting to about 25 percent at an ammonia-air ratio of approximately 0.05. This reduction in fuel flow resulted from the fact that approximately 35 percent of the injected ammonia burned in the engine combustion chamber at high ammonia-air ratios, as indicated in figure 15, and an even greater portion was burned at the low ammonia flows. Calculations to find the percentage of ammonia burned were based on an assumed 96 percent combustion efficiency for the main engine fuel.

The effects of ammonia on the compressor-outlet and turbine-outlet temperature profiles are presented in figure 16. Injection of the ammonia resulted in a slight flattening of the turbine-outlet temperature profiles (fig. 16(a)); but at constant average temperature of 1150° F (1610° R), the profiles with ammonia did not exceed the manufacturer's recommended limit any more than did the profiles with no injection. Because the ammonia and air were well mixed by the time they reached the compressor outlet, ammonia had little effect on compressor-outlet temperature profiles (fig. 16(b)), except for the reduction in level due to cooling.

Operating Experience

The engine ran smoothly with no loss of control over the range of conditions encountered. At low inlet-air temperatures, however, the ammonia flow was limited by engine blow-out to ammonia-air ratios not exceeding 0.046.

No detrimental effects on the engine components were observed during approximately 8 hours of operation with ammonia being injected into the engine inlet, during which time 27,000 pounds of ammonia were consumed. Some difficulty was encountered with oil congealing in lines exposed to inlet-air stream because of the low temperature encountered with ammonia injection. Corrective measures were taken and no further congealing was encountered.

CONCLUDING REMARKS

The results of this investigation indicate that ammonia is a promising coolant for use in obtaining a substantial amount of thrust augmentation at altitude flight conditions. Maximum net thrust augmentation obtained with ammonia injection at an altitude of 35,000 feet and a Mach number of 1.0 was 13 percent at the standard inlet-air temperature of 13° F, and 29 percent with an inlet-air temperature of 80° F.

At the same conditions, maximum augmentation with interstage water-alcohol injection (reference 3) was less than half that obtained with ammonia injection.

The jet thrust at high ammonia flows was 1 to 3 percent below the value calculated from engine performance characteristics based solely on the drop in inlet-air temperature due to evaporation of the ammonia. This difference was attributed to a nonuniform inlet-air temperature distribution, which resulted in a smaller increase in actual inlet weight flow and compressor pressure ratio than those calculated. In order to obtain maximum thrust augmentation it is therefore important to provide a uniform distribution of ammonia ahead of the engine inlet.

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APPENDIX - METHODS OF CALCULATION

The following symbols are used in this report:

A	area, sq ft
F	thrust, lb
g	acceleration due to gravity, 32.17 ft/sec^2
H	enthalpy, Btu/lb
h_c	lower heating value, Btu/lb
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
R	gas constant, ft-lb/(lb)($^{\circ}$R)
T	total temperature, $^{\circ}$R
V	velocity, ft/sec
v	specific volume, cu ft/lb
W	weight flow, lb/sec
γ	ratio of specific heats
η_f	$\dfrac{\text{actual heat released by fuel}}{\text{heating value of liquid fuel supplied}}$ = percentage of fuel burned
η_{NH_3}	$\dfrac{\text{actual heat released by ammonia}}{\text{heating value of liquid ammonia supplied}}$ = percentage of ammonia burned

2608

Subscripts:

a	air
f	fuel
g	gas
j	jet
NH_3	ammonia

n net

R reference condition

t exhaust-nozzle throat

0 free-stream condition

2608 Numbered subscripts refer to instrumentation stations within the engine (fig. 1). Prime superscripts refer to calculations assuming an isentropic process.

Air flow. - Air flow was determined from the measurements of temperature and pressure of station 1 as follows:

$$W_{a,1} = \frac{P_1 A_1}{T_1} \sqrt{\frac{2g\gamma T_1}{R(\gamma-1)} \left[\left(\frac{p_1}{P_1}\right)^{\frac{2}{\gamma}} - \left(\frac{p_1}{P_1}\right)^{\frac{\gamma+1}{\gamma}} \right]}$$

where $\gamma = 1.4$.

Jet thrust. - The jet thrust was determined from the following equation, which gives thrust for a 100 percent efficient convergent nozzle:

$$F_j = \frac{W_{g,8} V'_t}{g} + A_t (p'_t - p_0)$$

Net thrust. - The net thrust was determined by subtracting the inlet momentum of the air, at the particular simulated flight speed, from the jet thrust. Thus

$$F_n = F_j - \frac{W_a V_0}{g}$$

Theoretical jet thrust, inlet weight flow, and compressor pressure ratio. - Theoretical jet thrust, inlet weight flow (air plus ammonia), and compressor pressure ratio were determined by first running the engine at rated conditions over a range of inlet temperatures. Plots were then made of experimental jet thrust, air flow, and compressor pressure ratio versus inlet-air temperature. The weight of ammonia per pound of air required to drop the inlet-air temperature from T_1 to T_2 was calculated by means of the following equation:

$$\frac{W_{NH_3}}{W_a} = \frac{(H_{a,1} - H_{a,2})}{(H_{NH_3,2} - H_{NH_3,1})}$$

Enthalpies of ammonia were determined from reference 4.

By means of the experimental plots, then, values of jet thrust, inlet weight flow, and compressor pressure ratio were determined for T_2, corresponding to the calculated ammonia-air ratio. However, since 2608
the density of ammonia vapor is only about half that of air, the weight of mixture of ammonia and air handled by the engine was not as great as air alone. Hence, it was necessary to correct values of inlet weight flow and jet thrust for this difference in density. The specific volume entering the engine was determined as follows:

$$v_{g,2} = \frac{v_{a,2} + \frac{W_{NH_3}}{W_a} v_{NH_3,2}}{1 + \frac{W_{NH_3}}{W_a}}$$

The ratio of the density of the mixture to air alone at T_2 is $\frac{v_{a,2}}{v_{g,2}}$. Values of inlet weight flow were then multiplied by this density ratio to obtain theoretical weight flow corrected for density. Since the engine speed and turbine-outlet temperature were held constant by means of the variable-area exhaust nozzle, the jet thrust was directly proportional to the weight flow. Therefore, values of jet thrust were also multiplied by this density ratio. Changes in γ were considered negligible.

Percentage of ammonia burned. - The energy balance across the engine is defined by the following equation:

$$\eta_f \frac{W_f}{W_a} h_{c,f} + \eta_{NH_3} \frac{W_{NH_3}}{W_a} h_{c,NH_3} = H_a \Big]_1^5 + \frac{W_f}{W_a} \frac{Am+B}{m+1} \Big]_{T_R}^5 + \frac{W_{NH_3}}{W_a} D \Big]_{T_R}^5$$

where $\frac{Am+B}{m+1}$ represents the difference between the enthalpy of the products of combustion of the fuel-air mixture and the enthalpy of the oxygen removed from the air by their formation, and D represents a term similar to $\frac{Am+B}{m+1}$ for the combustion of an ammonia-air mixture (reference 5). Solving the preceding equation for the percentage of ammonia burned η_{NH_3} results in the following equation:

$$\eta_{NH_3} = \frac{\left. H_a \right]_1^5 + \left. \frac{W_f}{W_a}\left(\frac{Am+B}{m+1}\right) \right]_{T_R}^5 + \left. \frac{W_{NH_3}}{W_a} D \right]_{T_R}^5 - \eta_f \frac{W_f}{W_a} h_{c,f}}{\frac{W_{NH_3}}{W_a} h_{c,NH_3}}$$

2608

A constant value of 0.96 was assumed for the percentage of fuel burned. The inlet temperature of the liquid ammonia and fuel supplied to the engine was assumed 540° R, which was taken as the reference temperature T_R.

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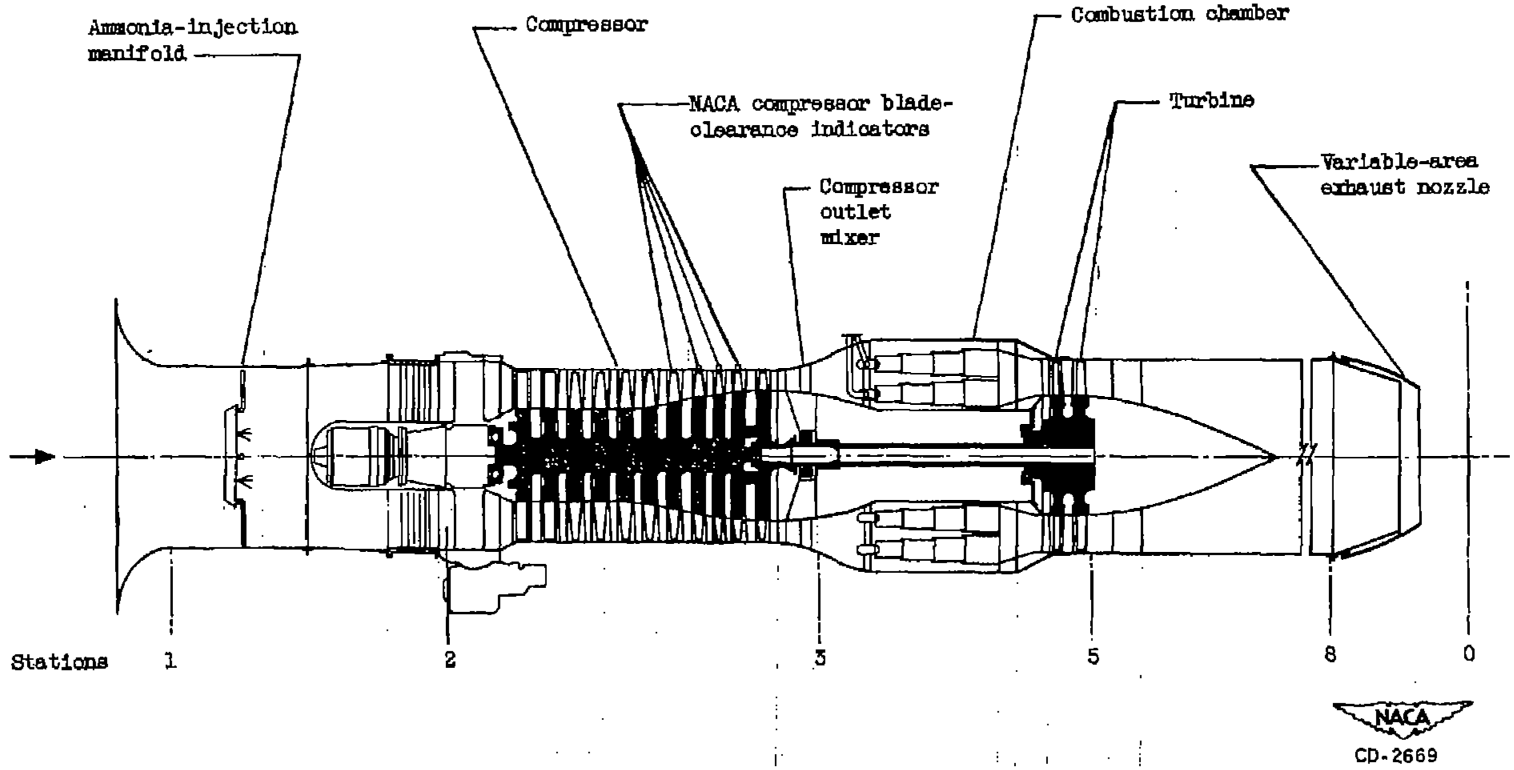

Figure 1. - Sectional view of engine showing instrumentation stations.

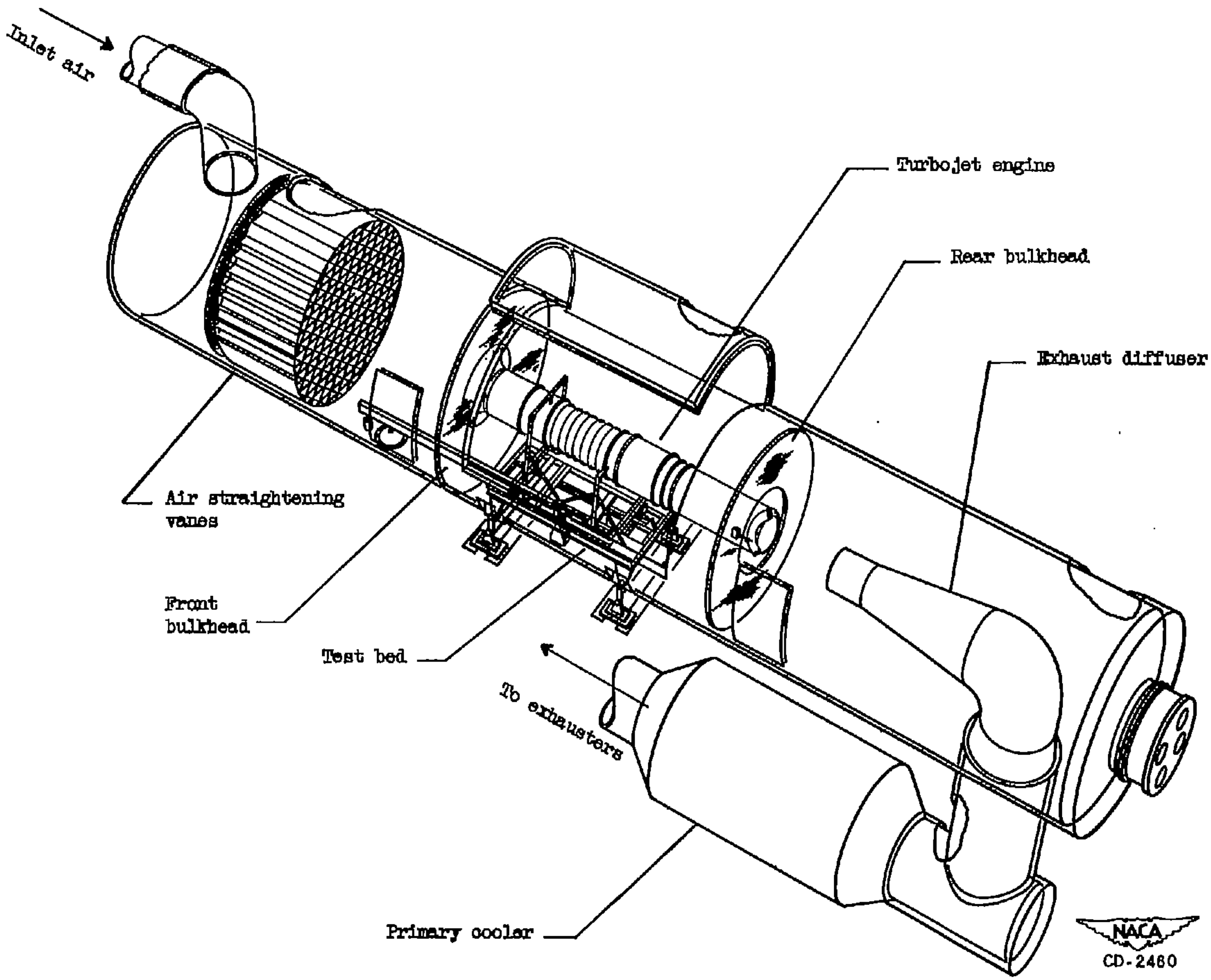

Figure 2. - Illustration of altitude test chamber showing turbojet engine installed on test bed.

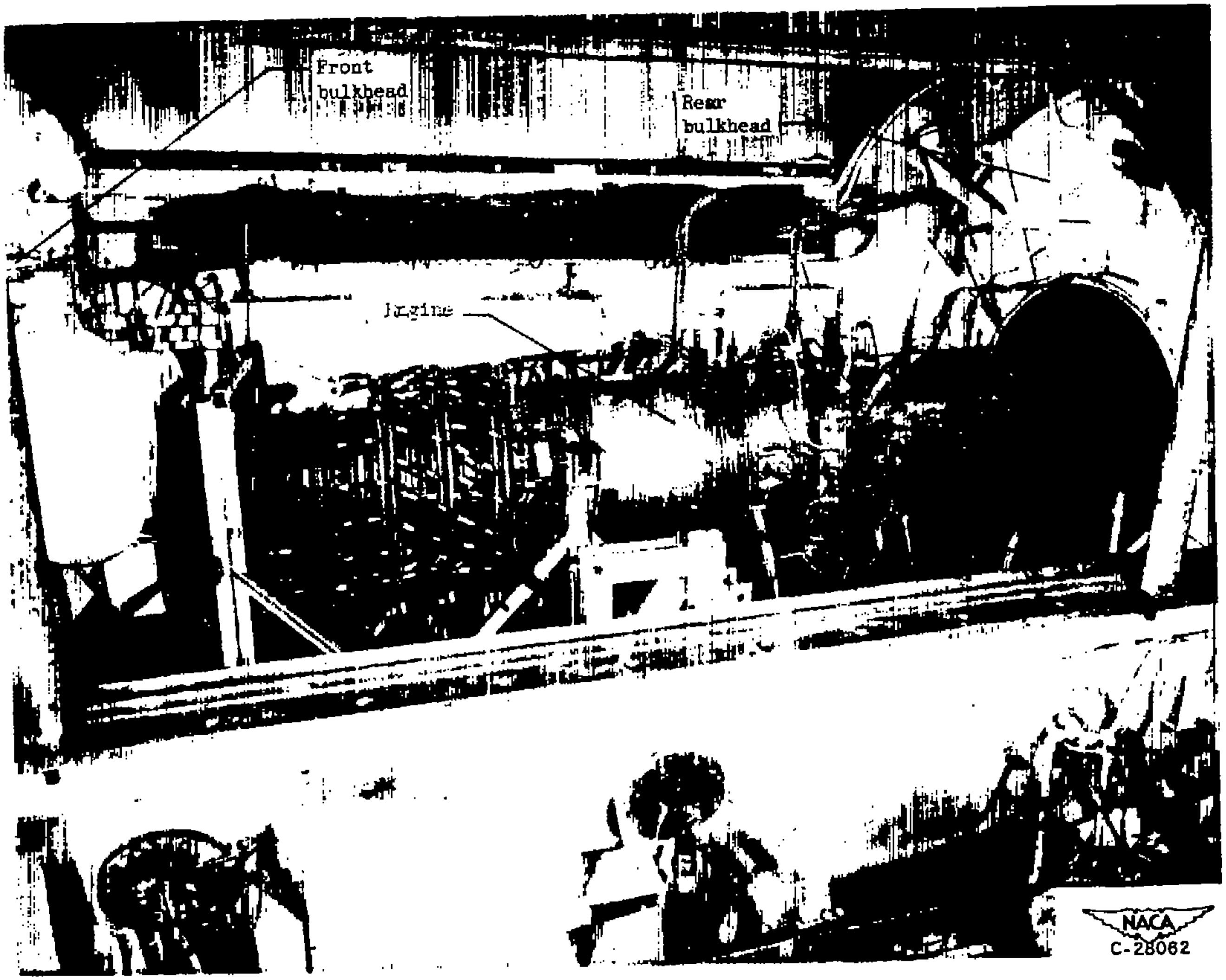

Figure 3. - Turbojet engine installed in altitude chamber.

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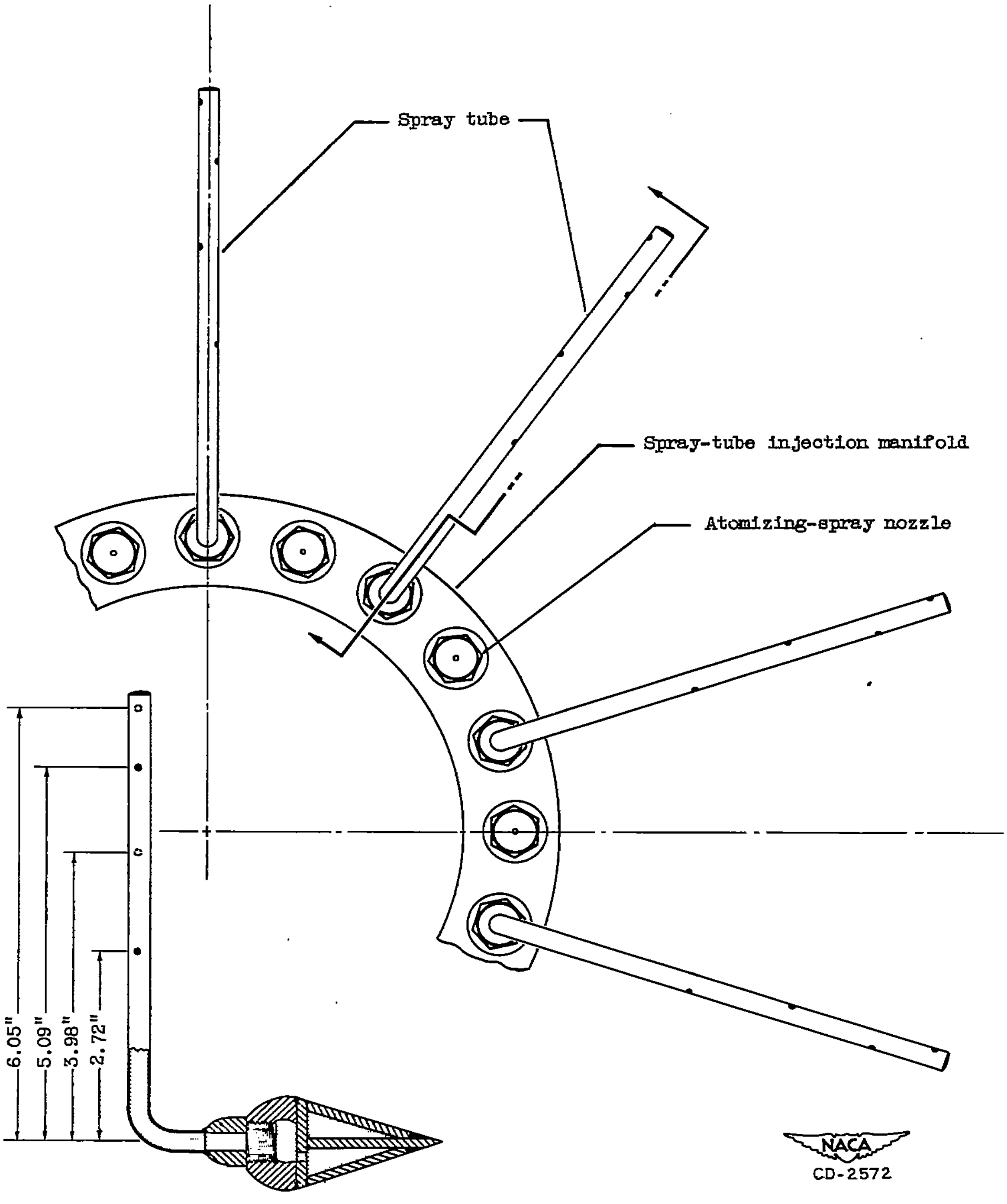

Figure 4. - Ammonia-injection manifold.

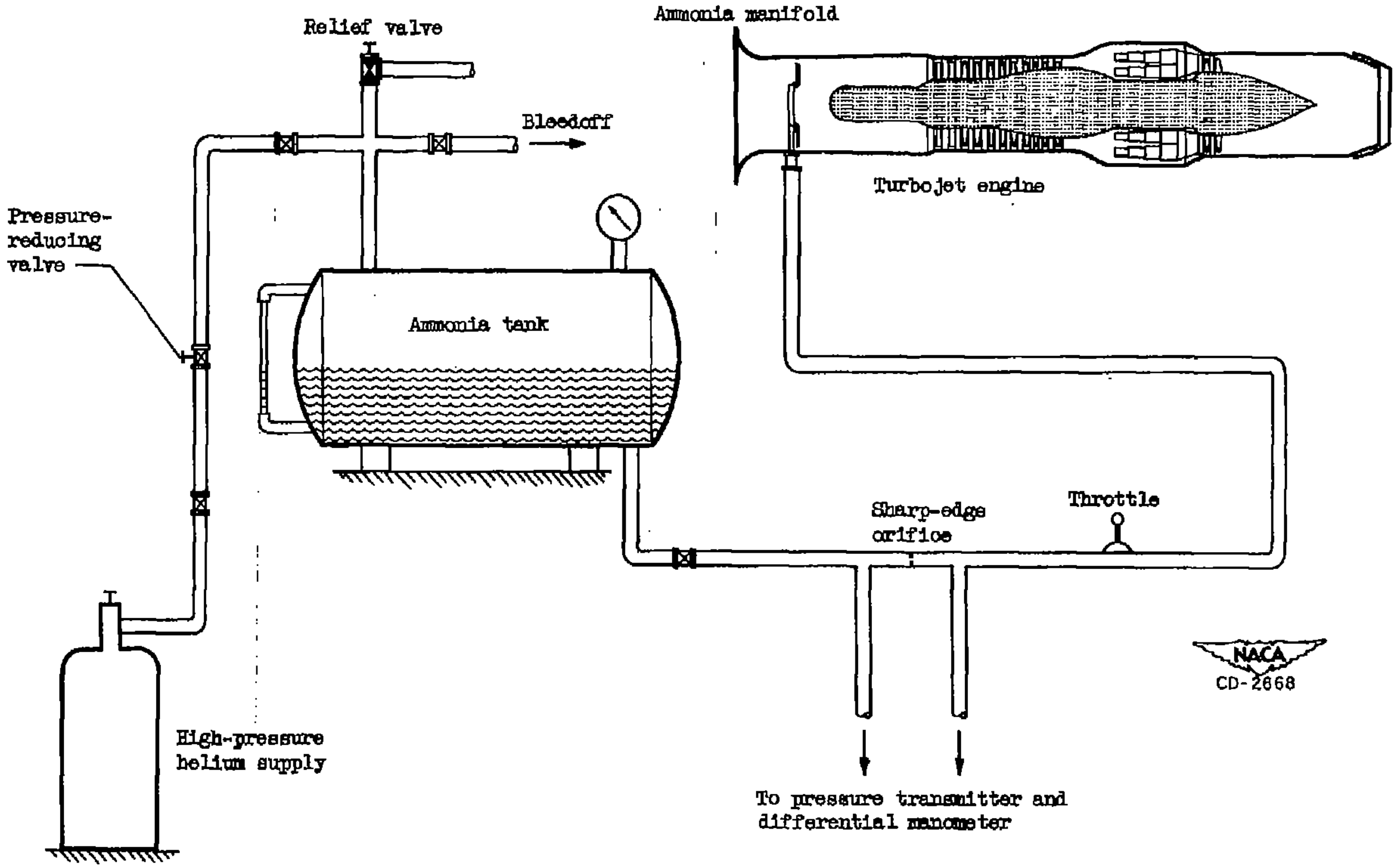

Figure 5. - Schematic diagram of ammonia-injection system.

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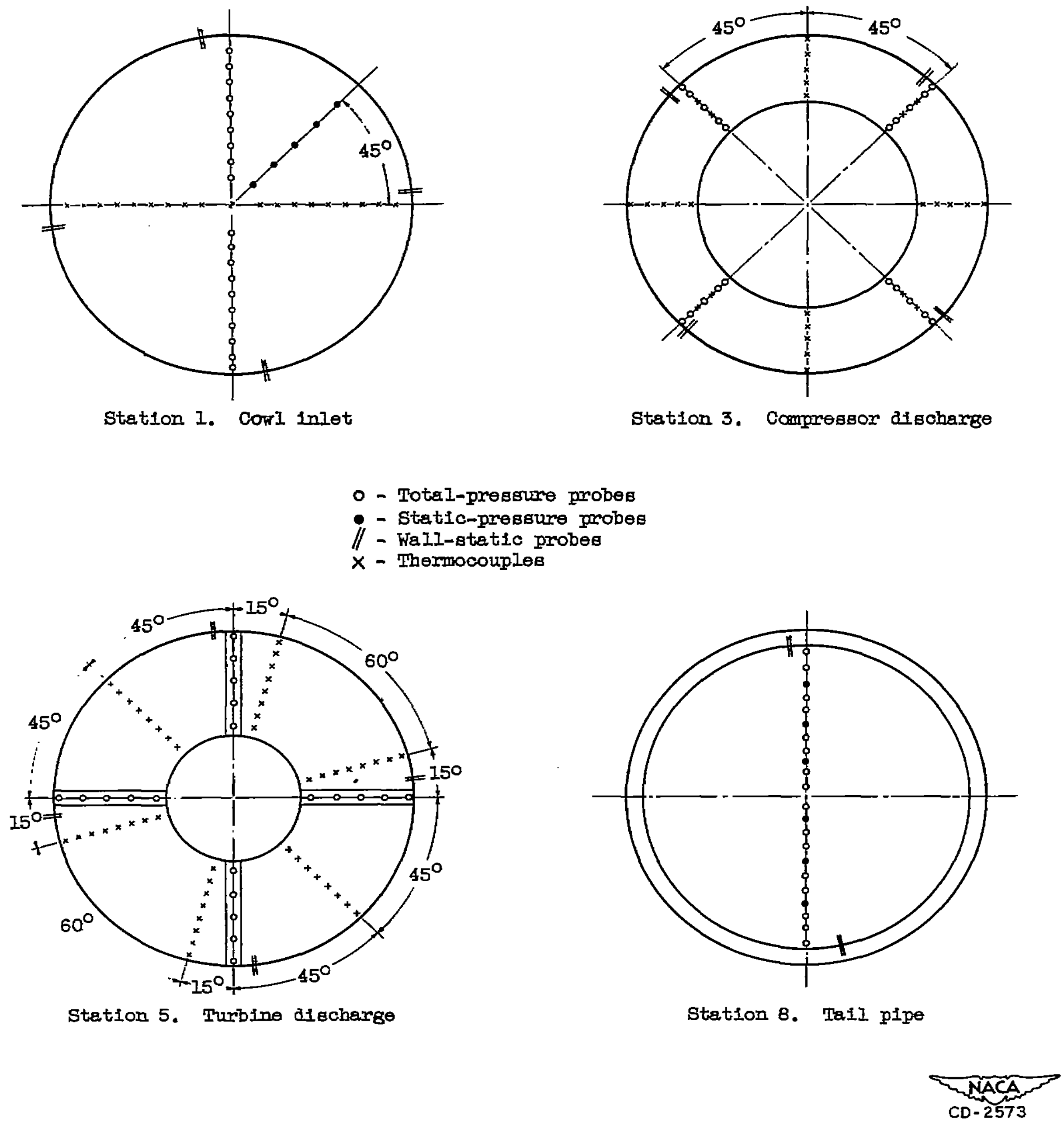

Figure 6. - Temperature and pressure instrumentation installed in the engine.

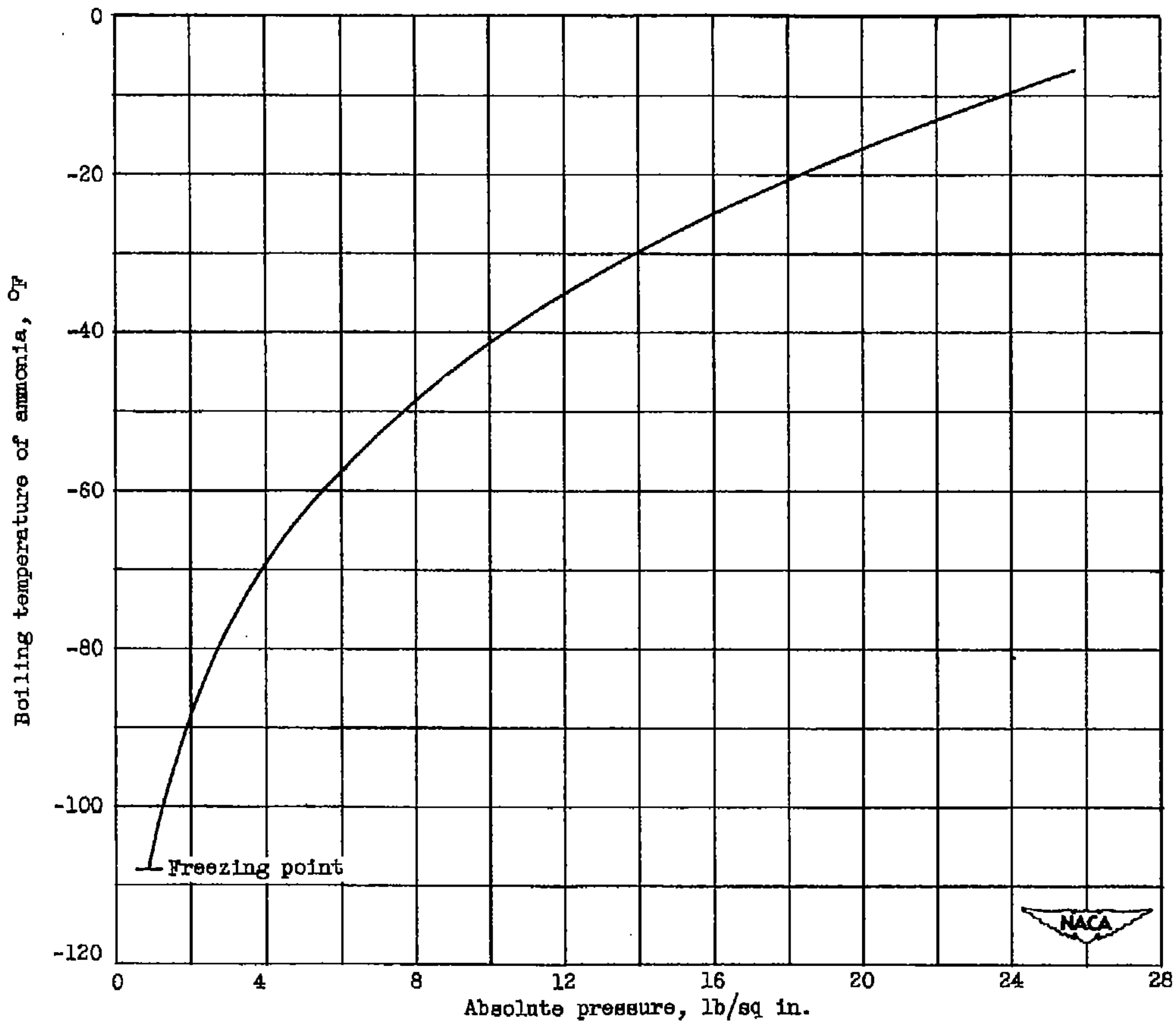

Figure 7. - Effect of absolute pressure on the boiling temperature of ammonia.

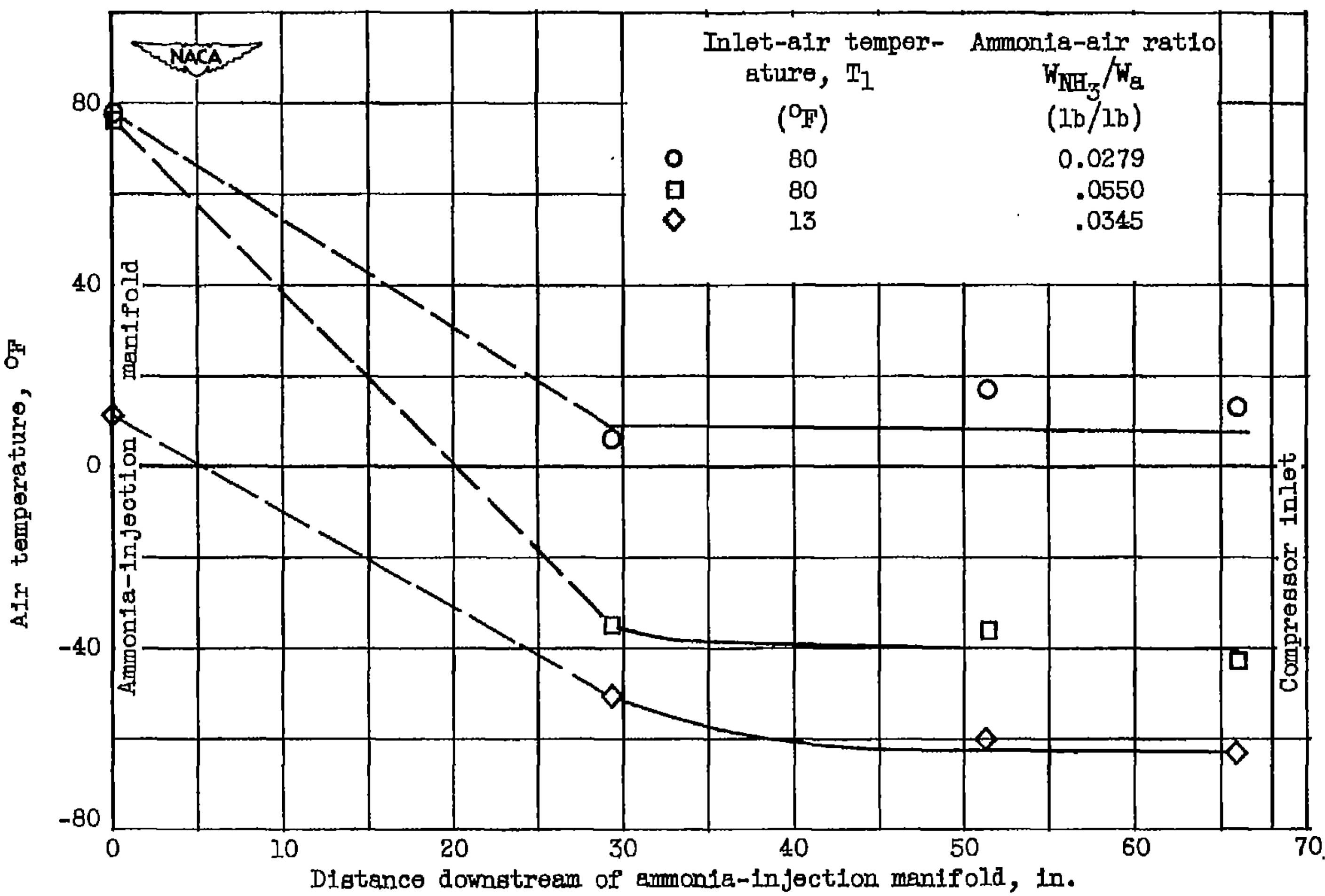

Figure 8. - Variation of air temperature with distance downstream of ammonia-injection manifold for several inlet-air temperatures and several ammonia-air ratios. Engine speed, 12,500 rpm; average turbine-outlet temperature, 1610° R; altitude, 35,000 feet; flight Mach number, 1.0.

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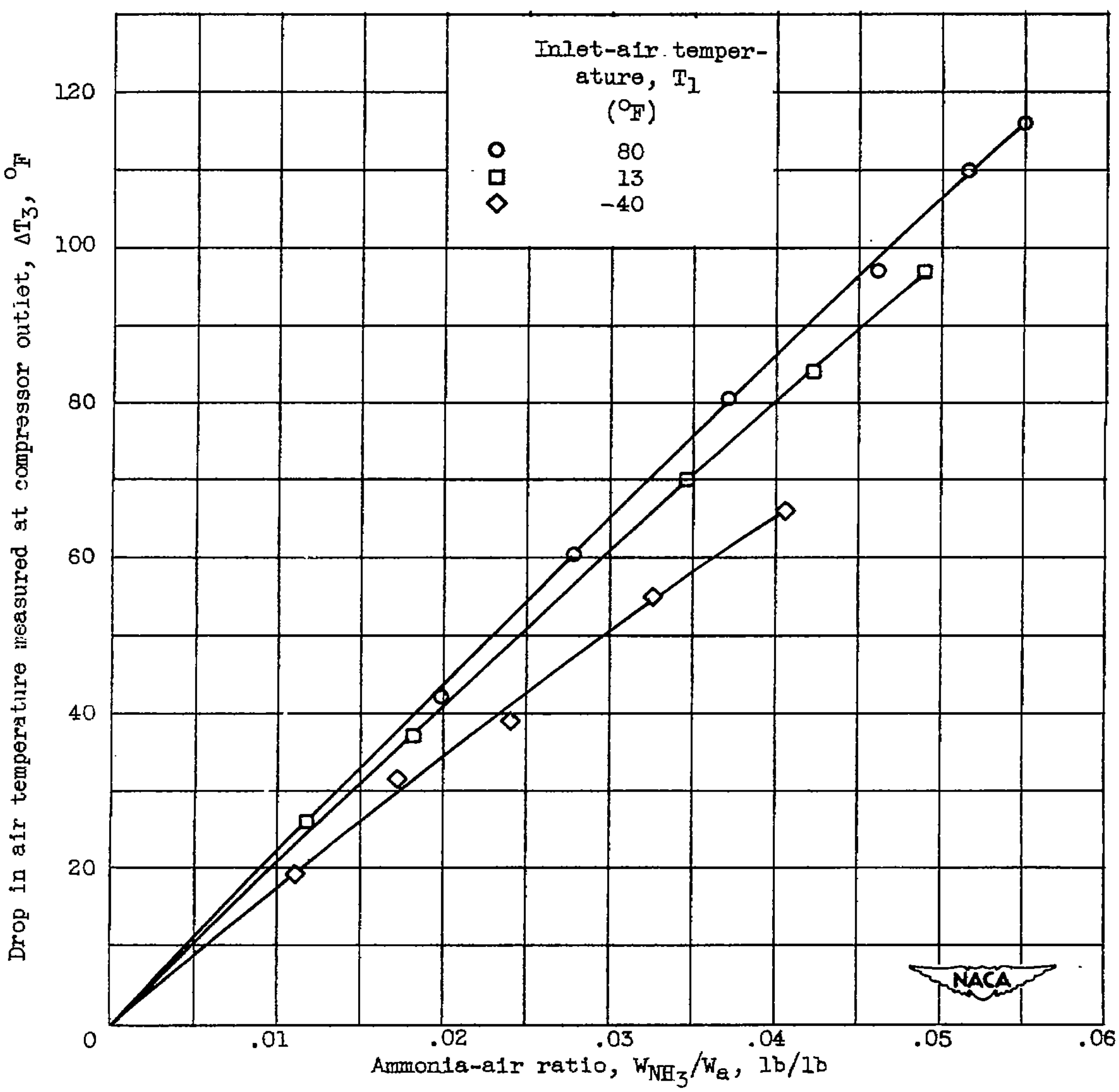

Figure 9. - Drop in air temperature measured at compressor outlet versus ammonia-air ratio for several inlet-air temperatures. Engine speed, 12,500 rpm; average turbine-outlet temperature, 1610° R; altitude, 35,000 feet; flight Mach number, 1.0.

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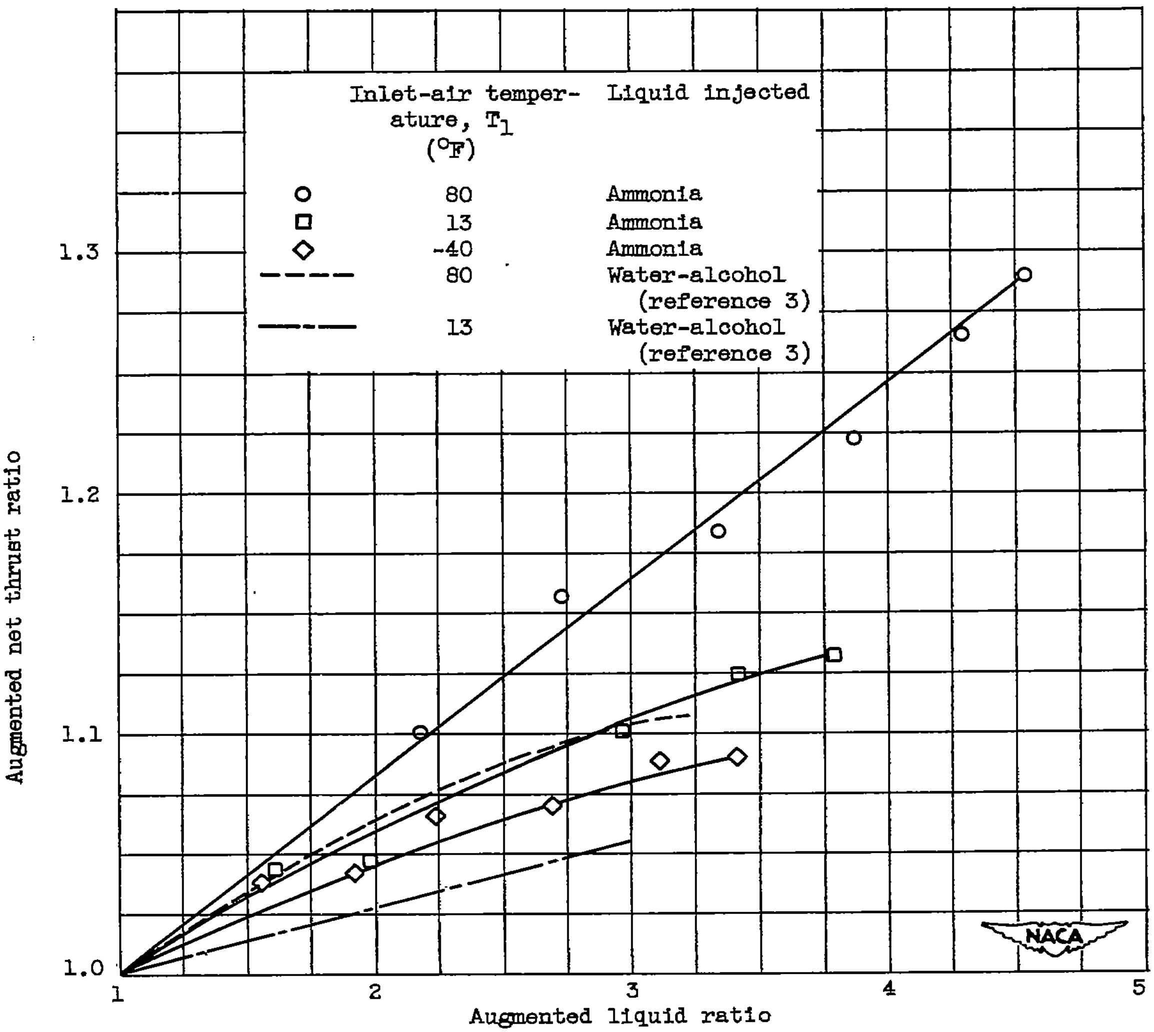

Figure 10. - Variation of augmented thrust ratio with augmented liquid ratio for several inlet-air temperatures. Engine speed, 12,500 rpm; average turbine-outlet temperature, 1610° R; altitude, 35,000 feet; flight Mach number, 1.0.

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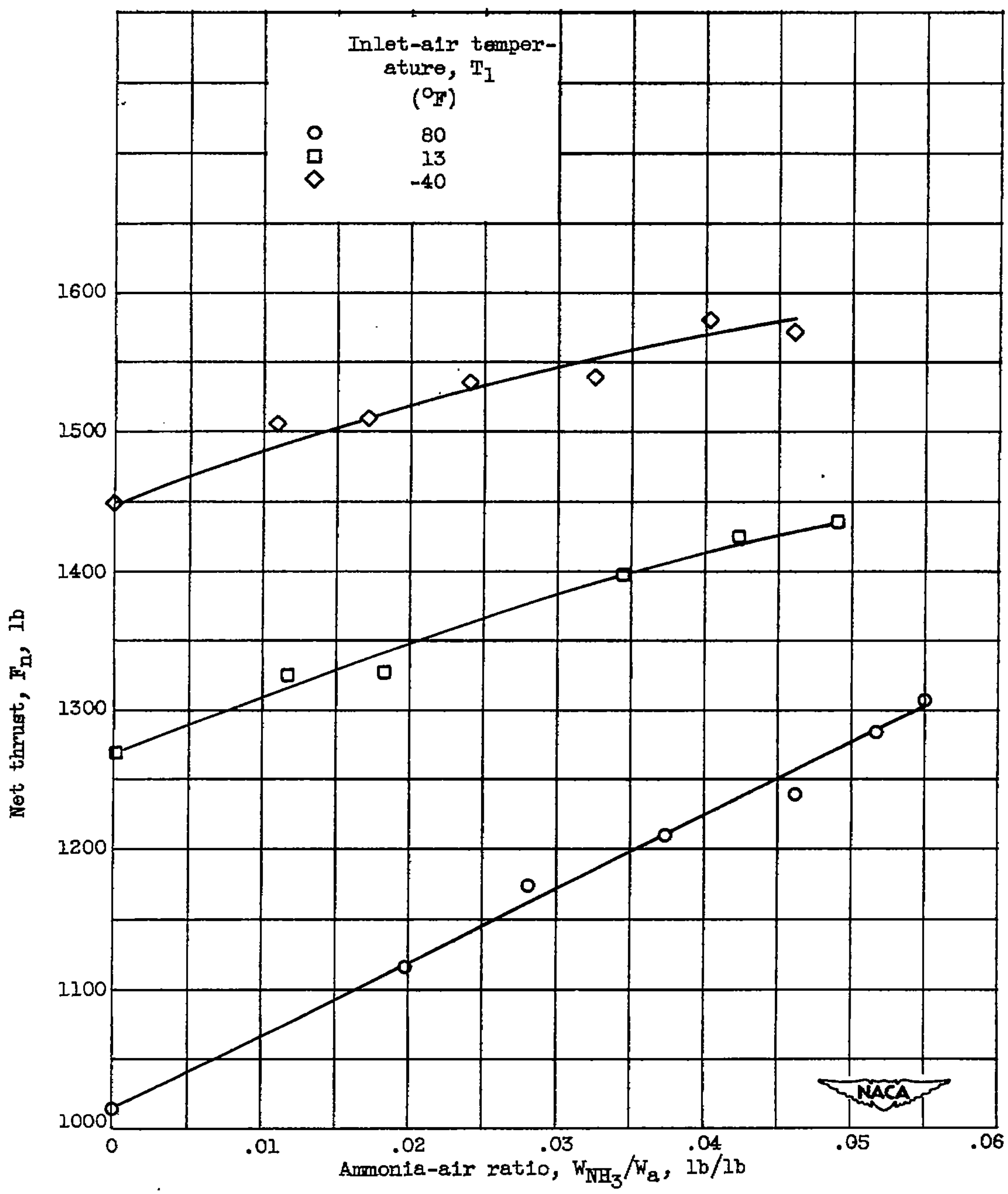
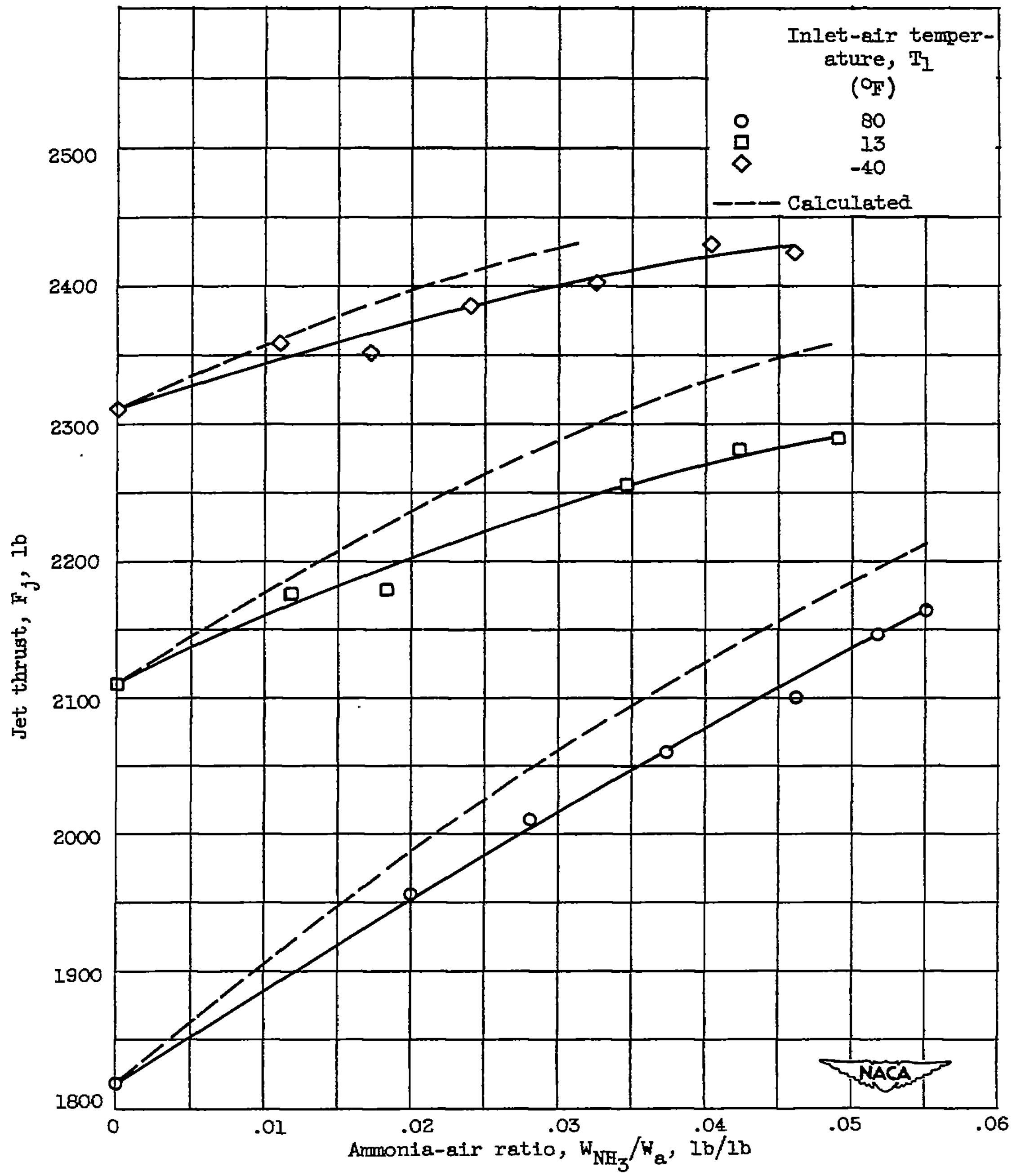


(a) Net thrust.

Figure 11. - Variation of thrust with ammonia-air ratio for several inlet-air temperatures. Engine speed, 12,500 rpm; average turbine-outlet temperature, 1610° R; altitude, 35,000 feet; flight Mach number, 1.0.



(b) Jet thrust.

Figure 11. - Concluded. Variation of thrust with ammonia-air ratio for several inlet-air temperatures. Engine speed, 12,500 rpm; average turbine-outlet temperature, 1610° R; altitude, 35,000 feet; flight Mach number, 1.0.

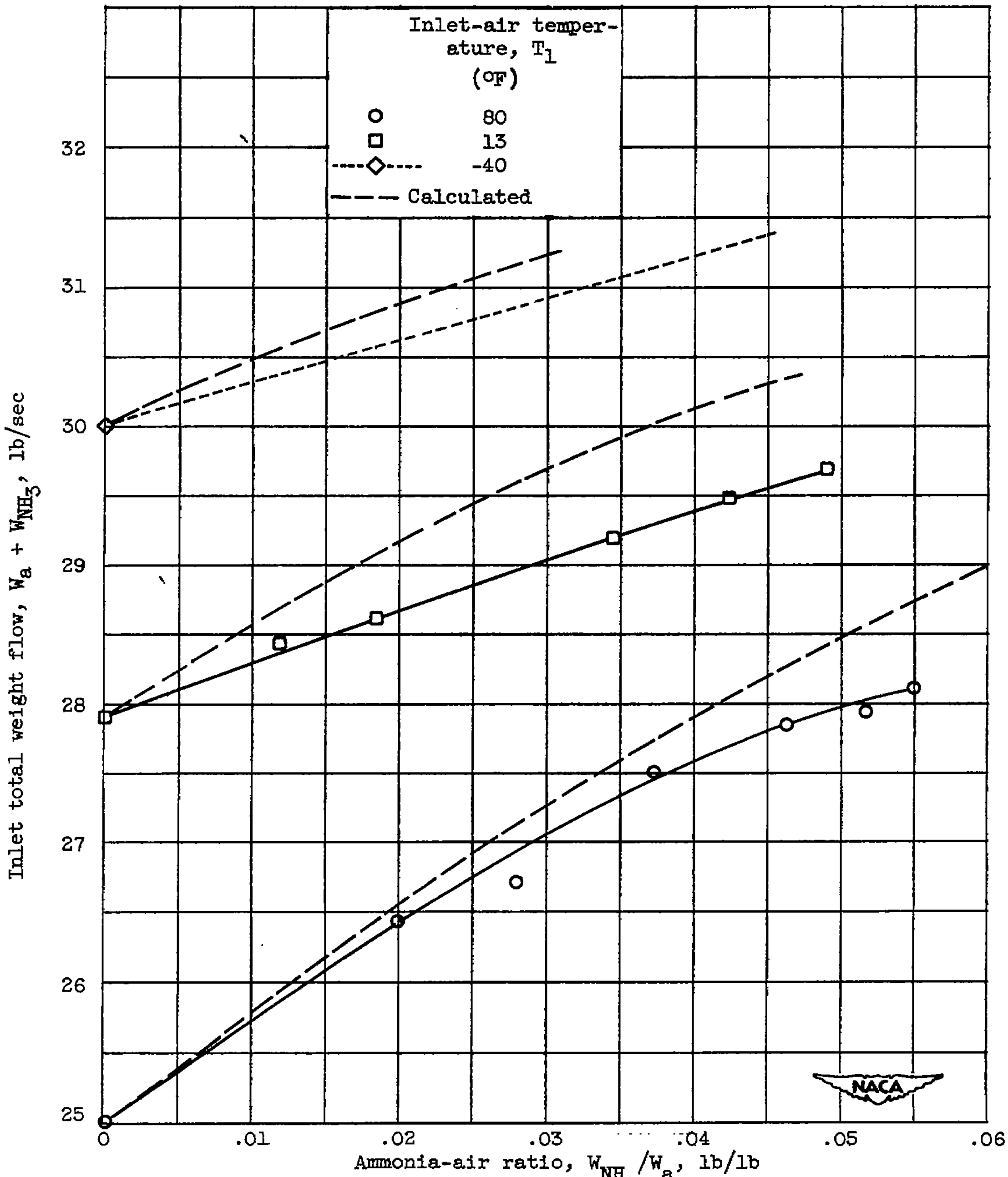


Figure 12. - Variation of inlet weight flow with ammonia-air ratio for several inlet-air temperatures. Engine speed, 12,500 rpm; average turbine-outlet temperature, 1610° R; altitude, 35,000 feet; flight Mach number, 1.0.

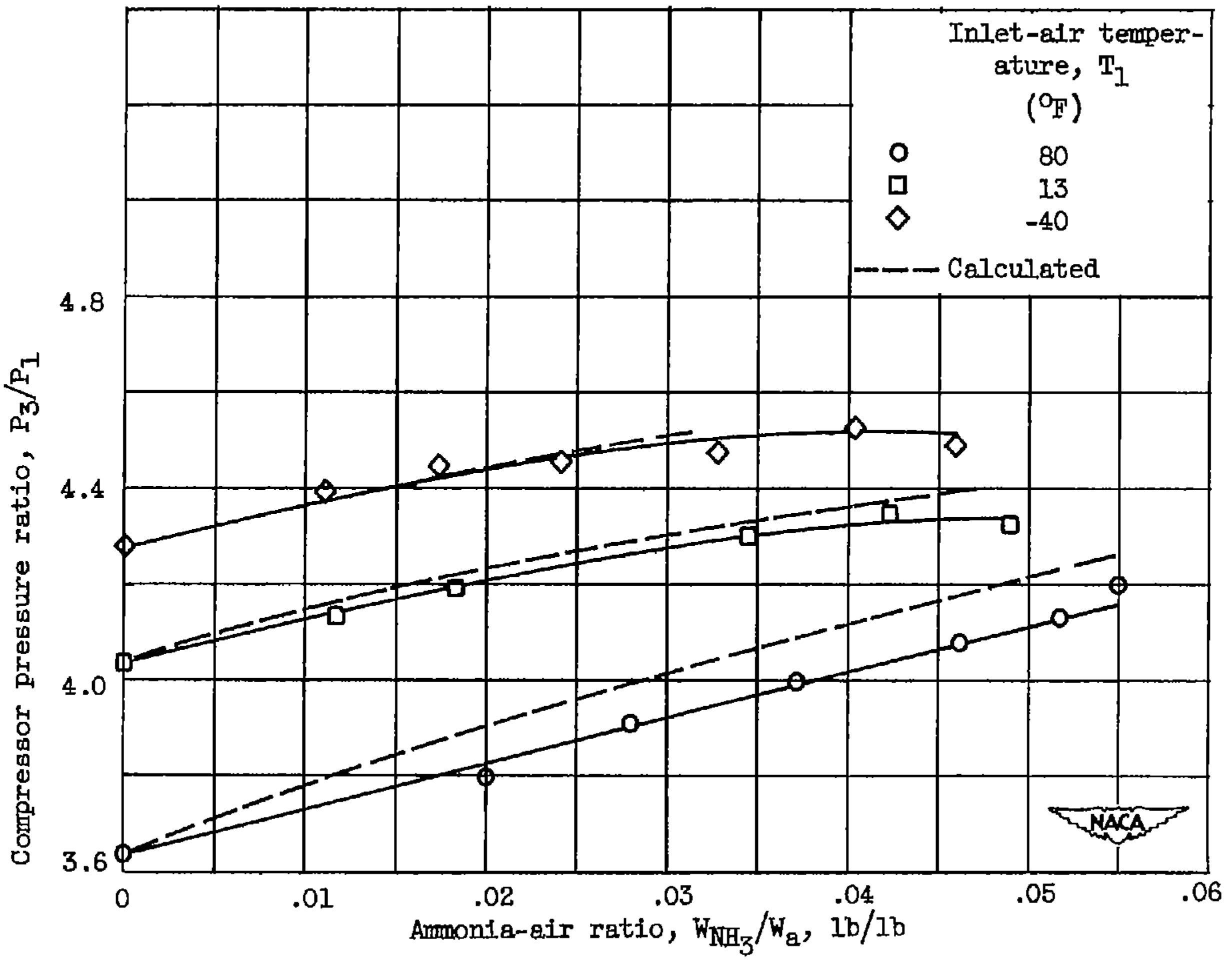


Figure 13. - Variation of compressor pressure ratio with ammonia-air ratio for several inlet-air temperatures. Engine speed, 12,500 rpm; average turbine-outlet temperature, 1610° R; altitude, 35,000 feet; flight Mach number, 1.0.

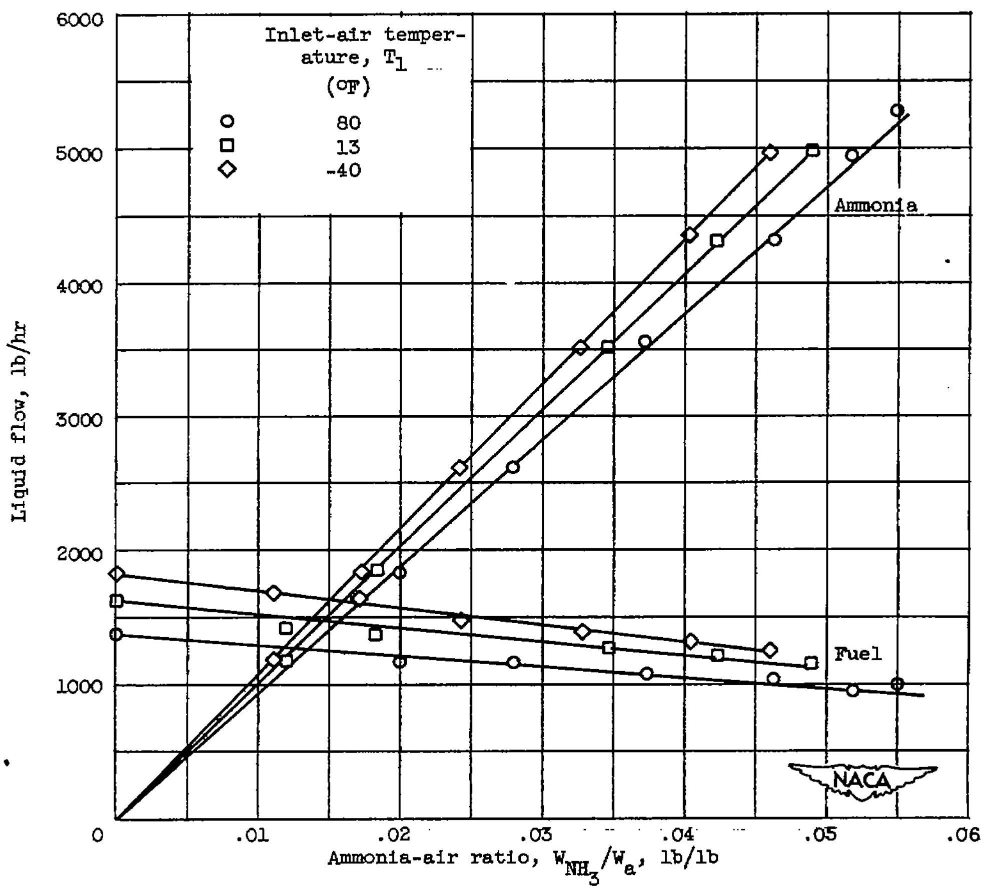


Figure 14. - Variation of ammonia flow and fuel flow with ammonia-air ratio for several inlet-air temperatures. Engine speed, 12,500 rpm; average turbine-outlet temperature, 1610° R; altitude, 35,000 feet; flight Mach number, 1.0.

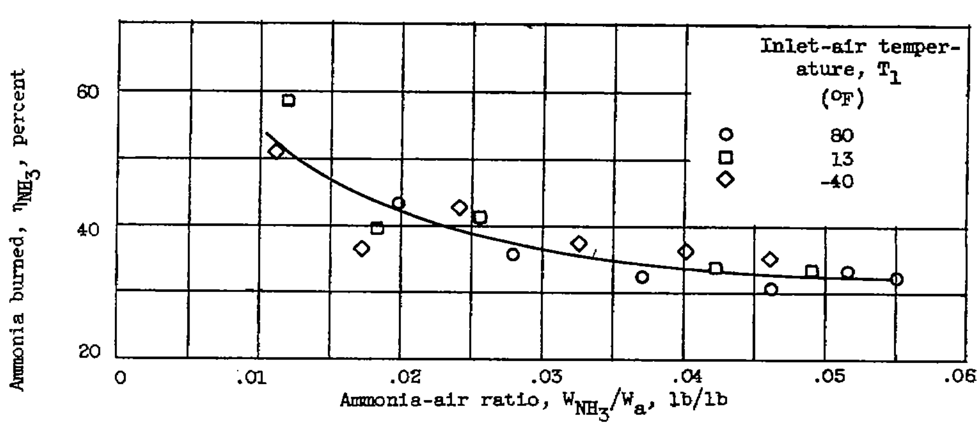
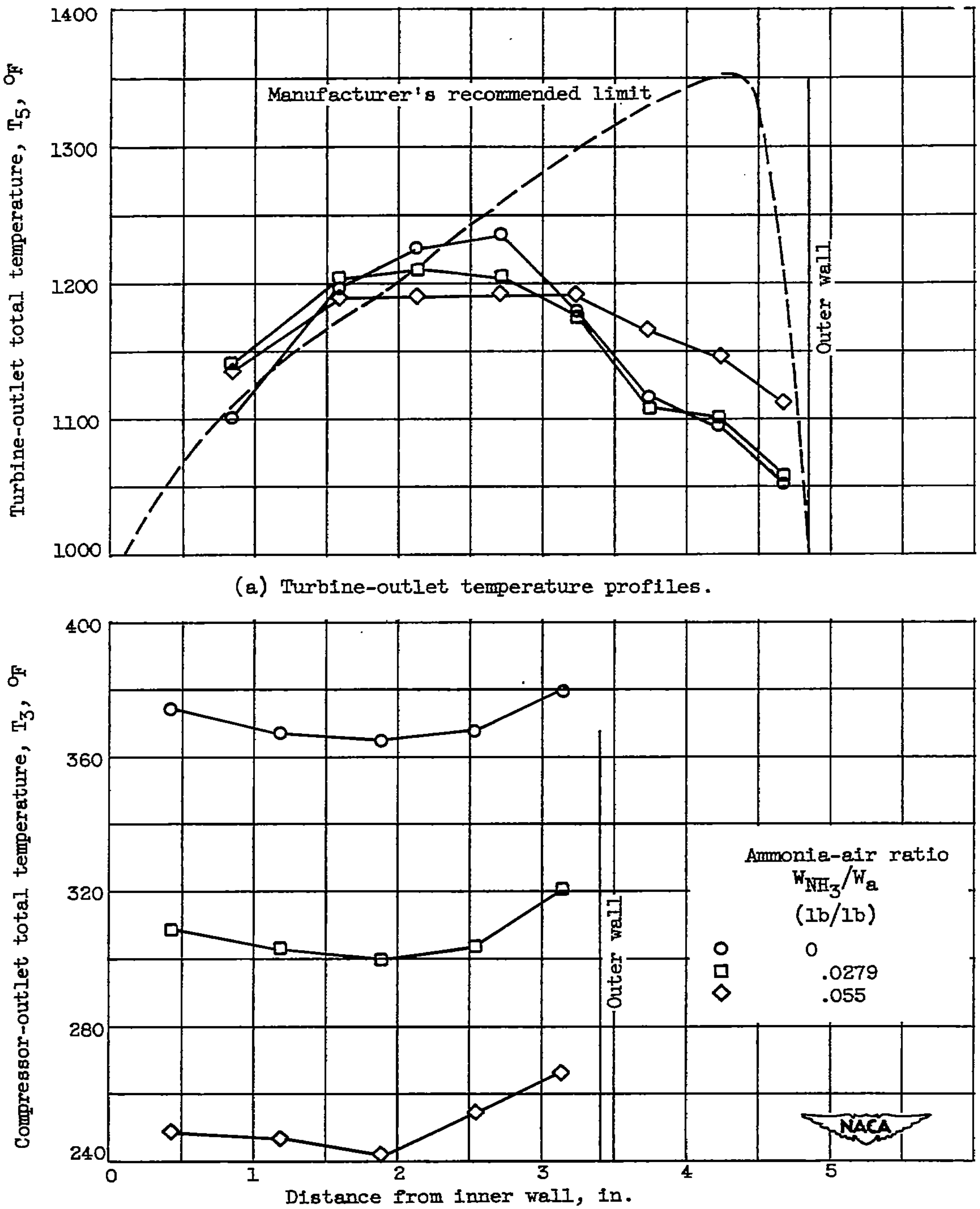


Figure 15. - Percentage of ammonia burned in engine combustion chamber versus ammonia-air ratio for several inlet-air temperatures. Engine speed, 12,500 rpm; average turbine-outlet temperature, 1610° R; altitude, 35,000 feet; flight Mach number, 1.0.



(a) Turbine-outlet temperature profiles.

(b) Compressor-outlet temperature profiles.

Figure 16. - Turbine- and compressor-outlet temperature profiles for several ammonia-air ratios. Inlet-air temperature, 80° F; engine speed, 12,500 rpm; average turbine-outlet temperature, 1610° R; altitude, 35,000 feet; flight Mach number, 1.0.

SECURITY INFORMATION

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